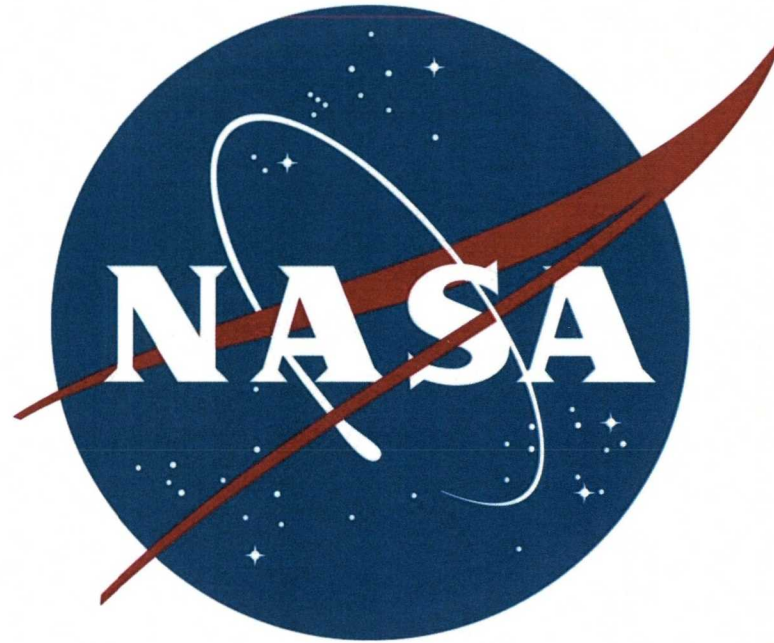
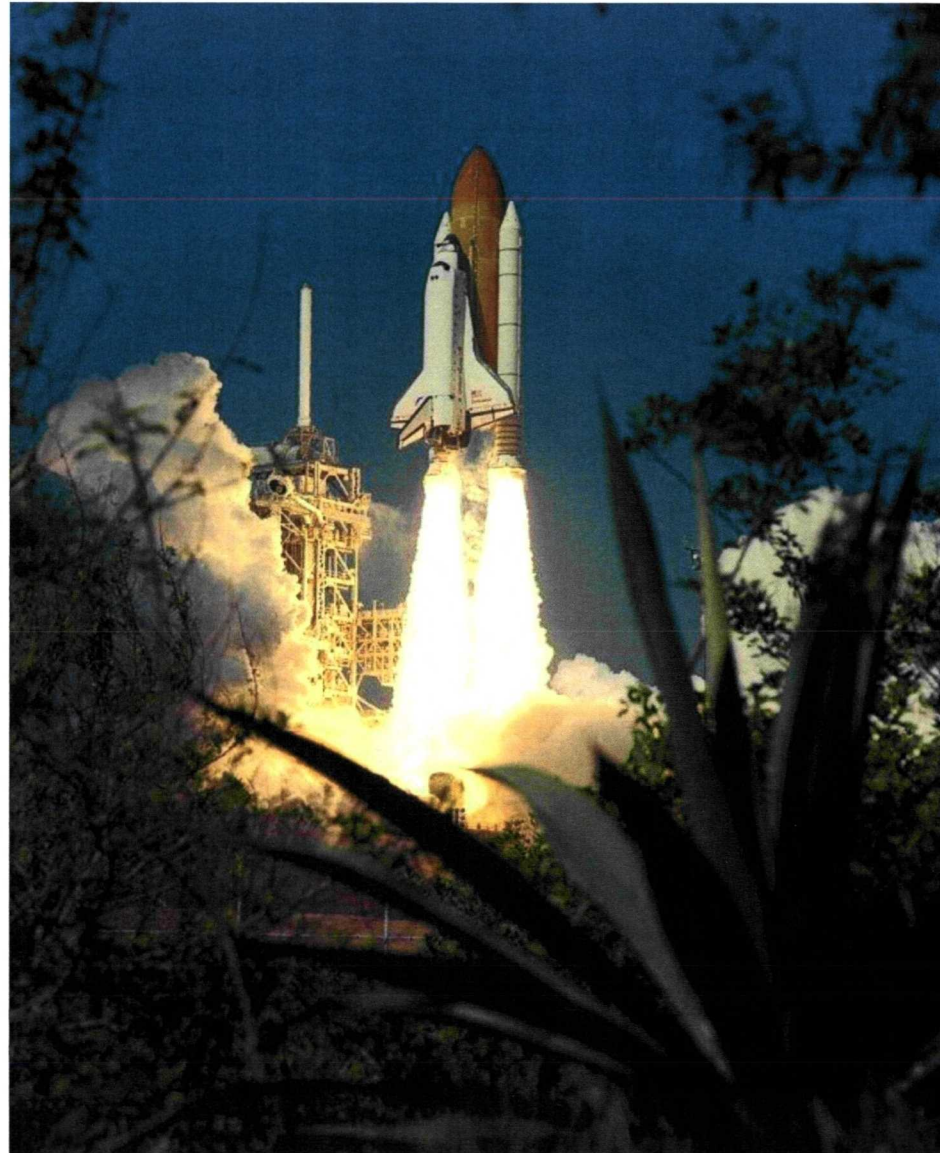


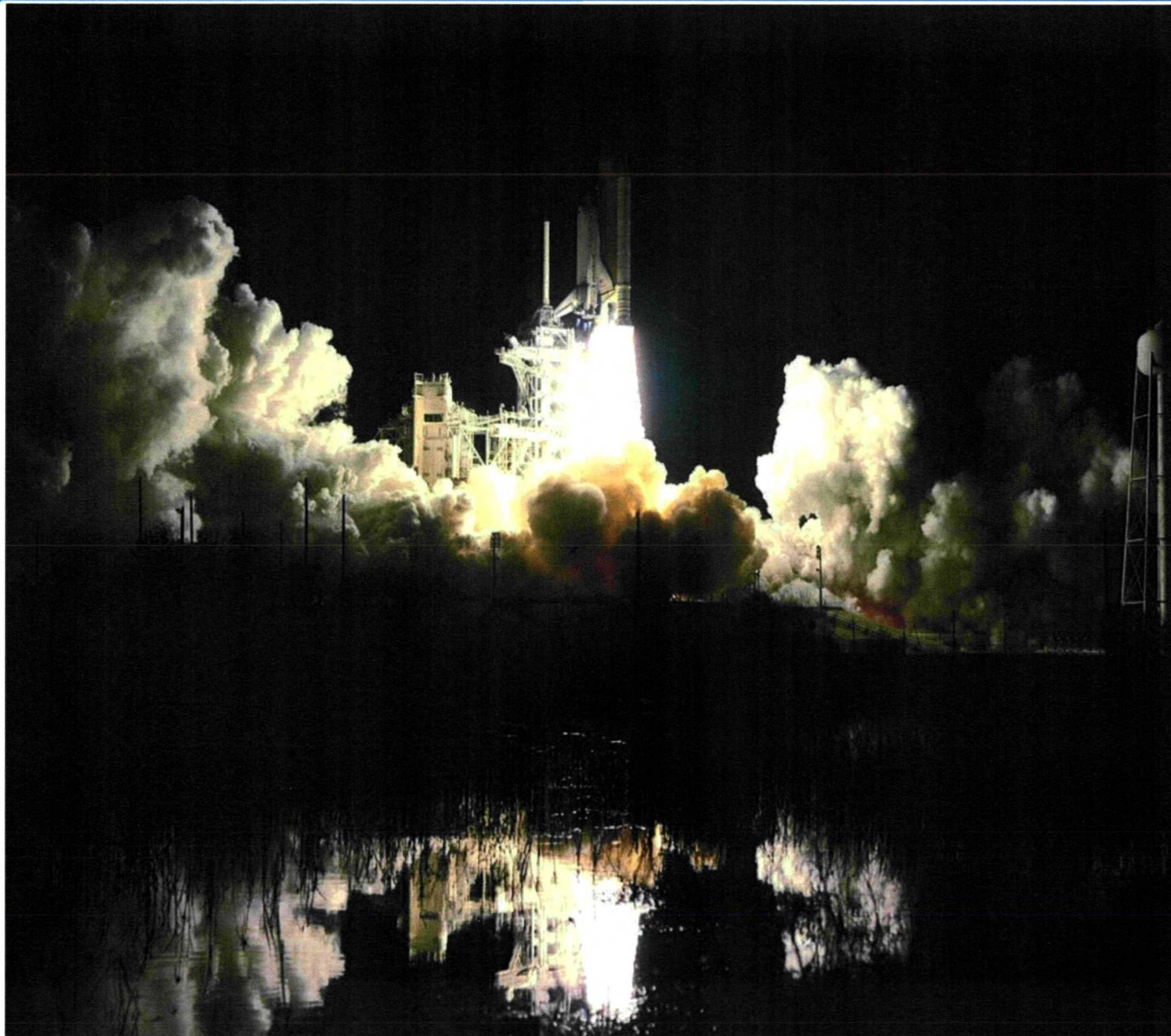
## Expendable Launch Vehicles Briefing



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August 2010



















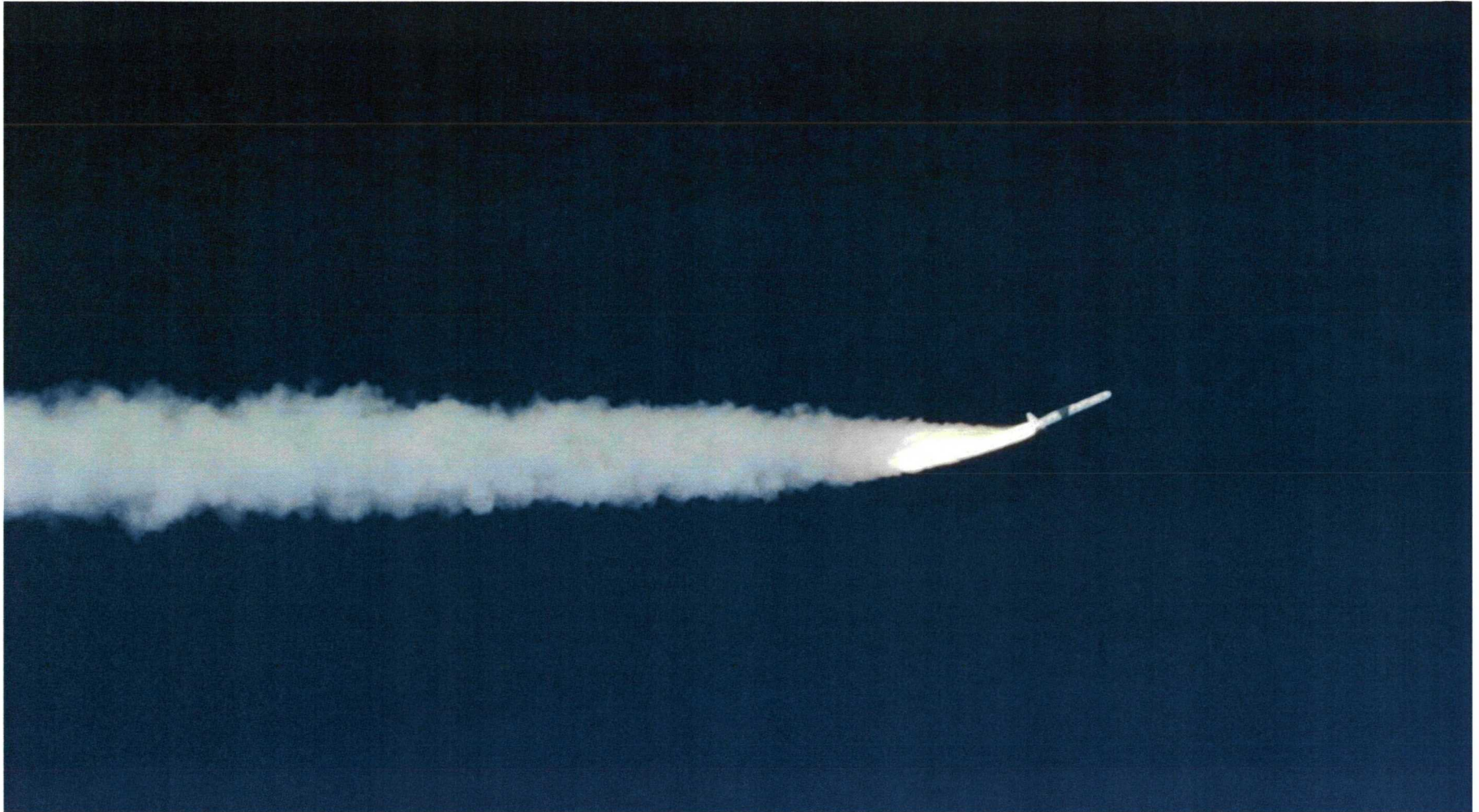




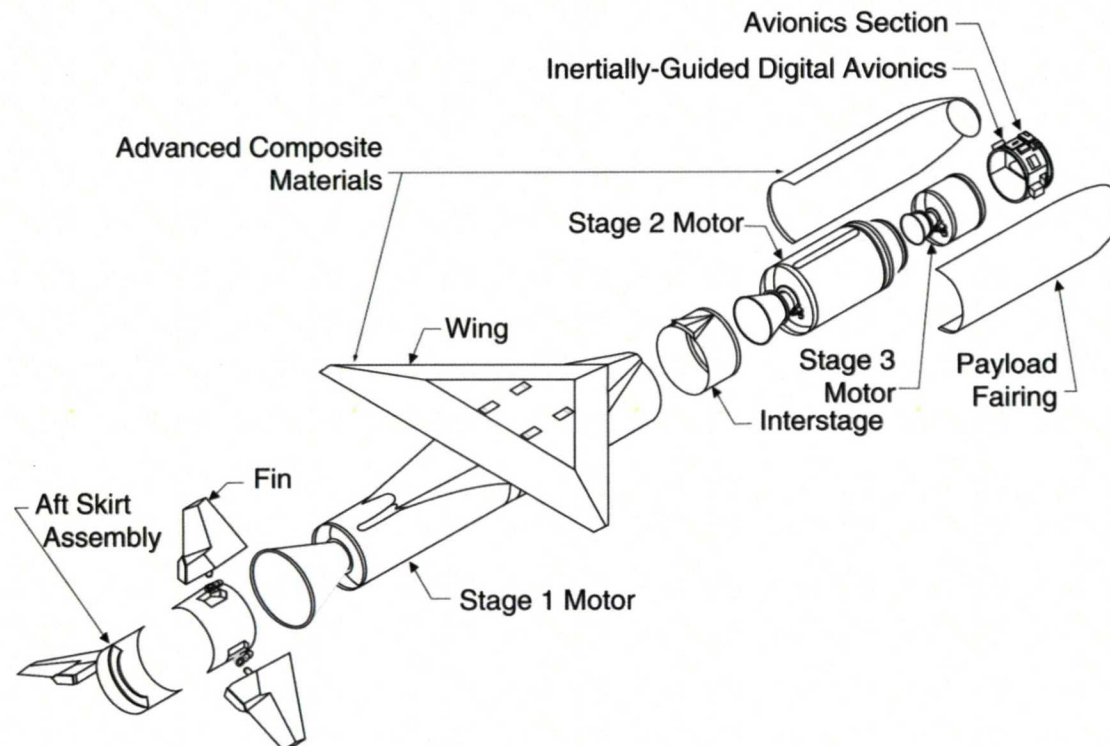
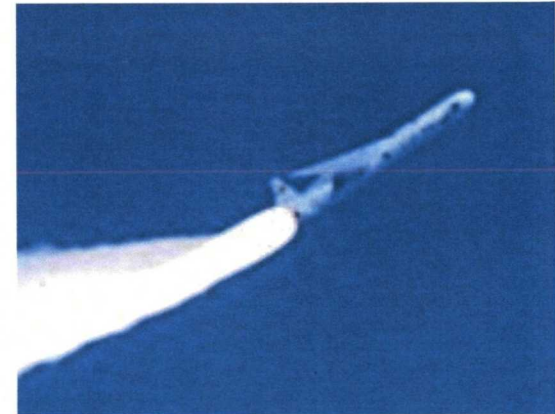


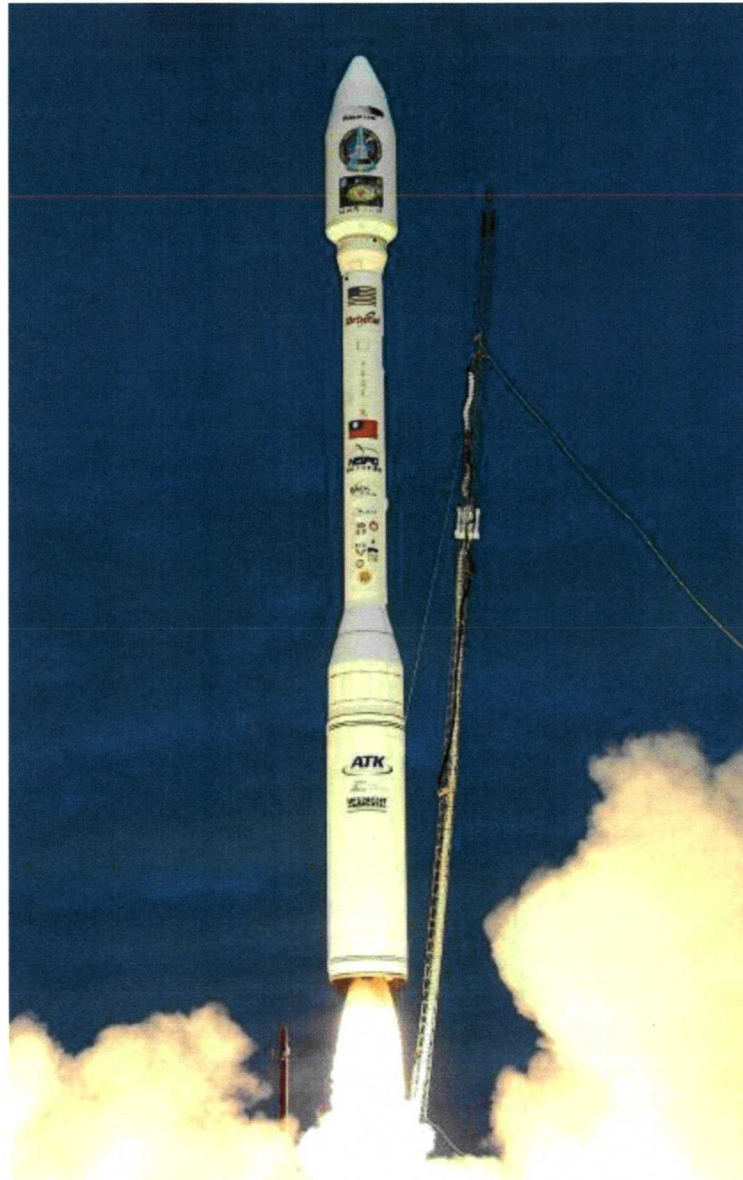




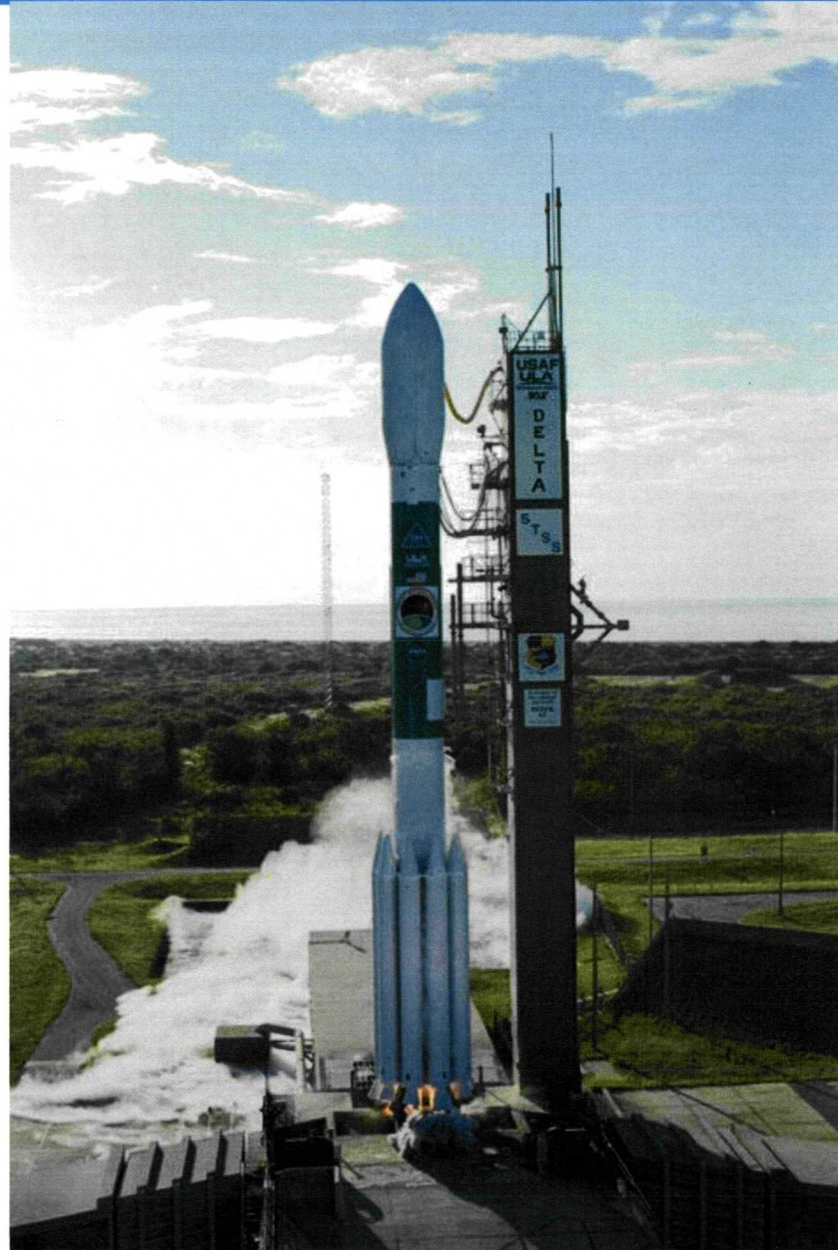






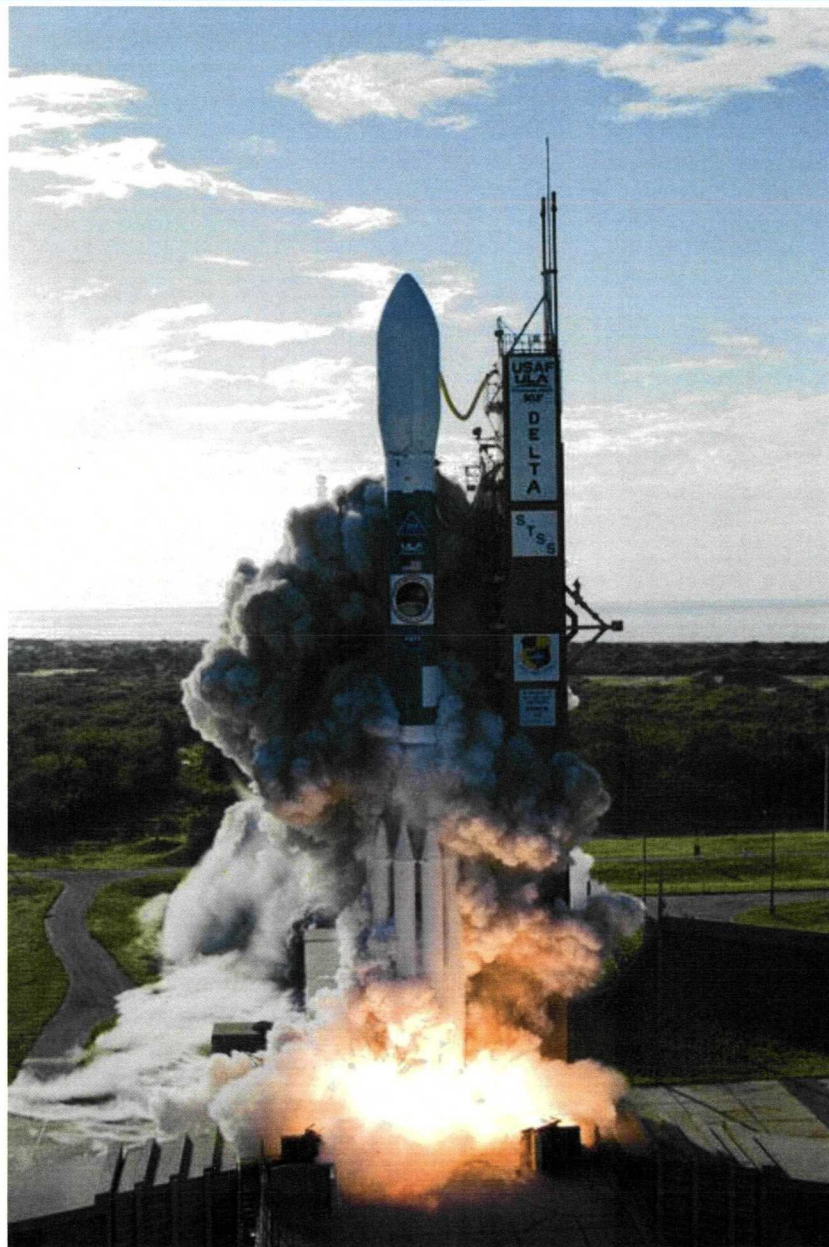












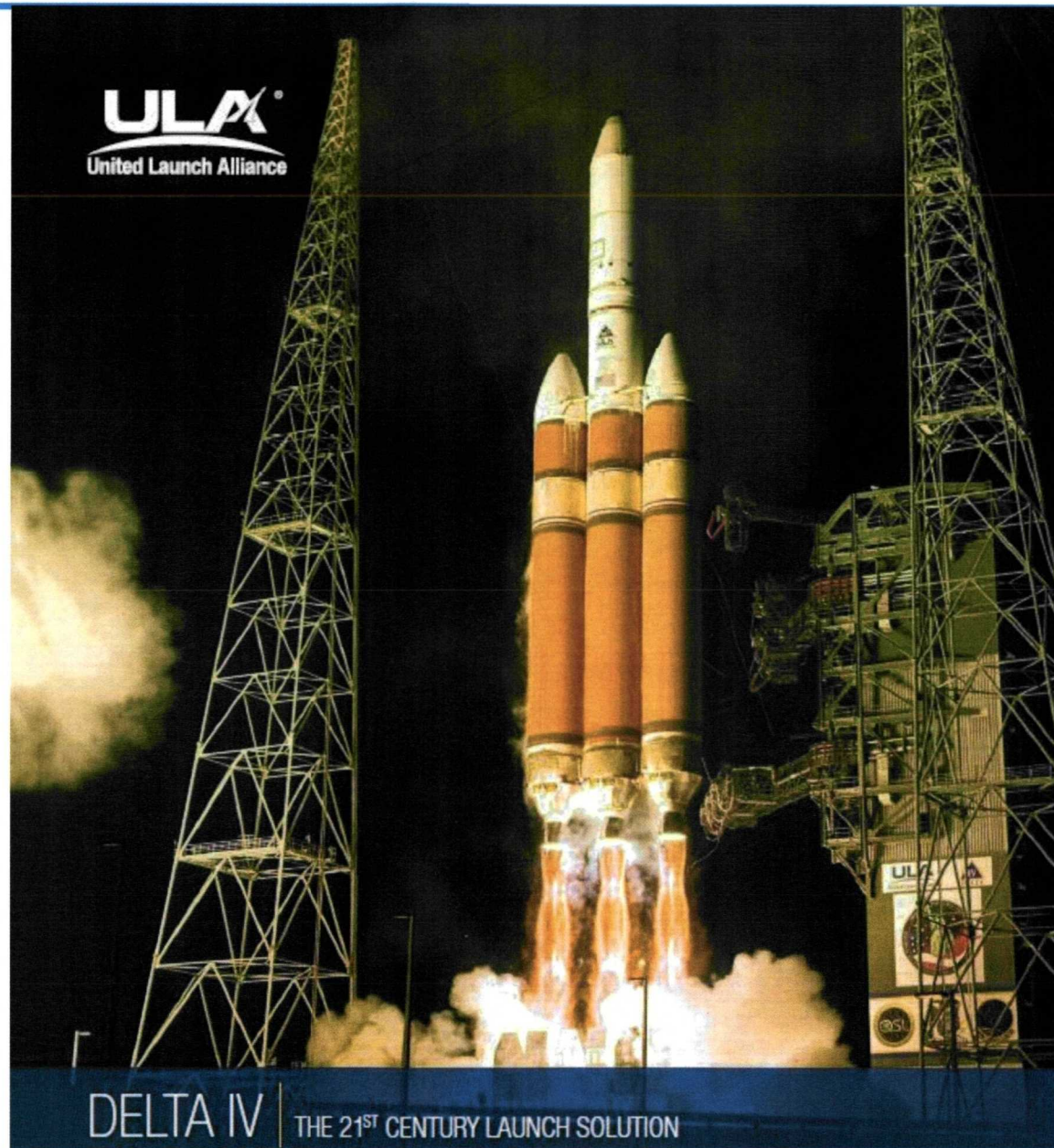




















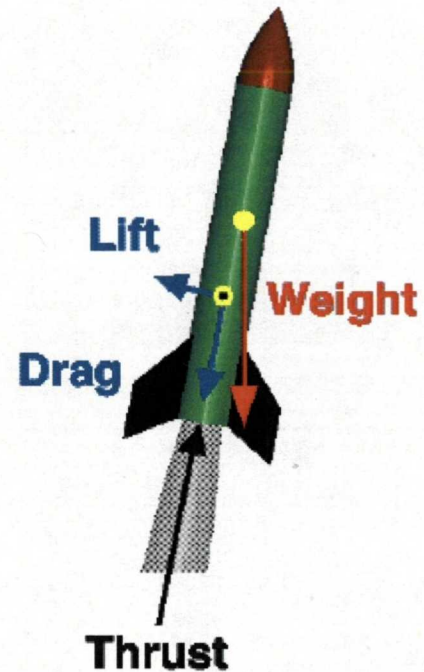
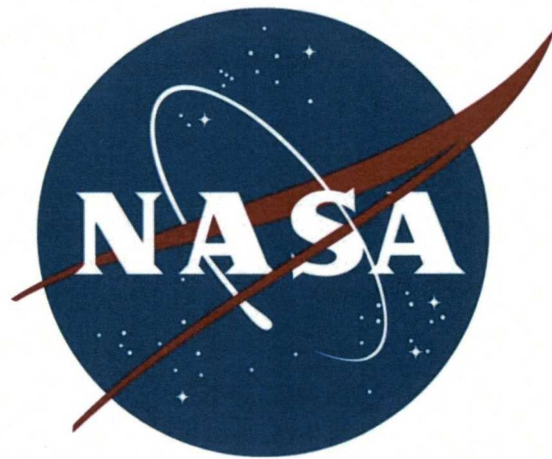
## References

All information came from publicly available information:  
NASA KSC Image Library  
Launch Service Provider's Mission Planning Guides and Websites





## Basic Rocketry Physics



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## Newton's Laws of Motion

---

Newton's Laws of Motion are three physical laws that form the basis for classical mechanics. They describe the relationship between the forces acting on a body and its motion due to those forces. They have been expressed in several different ways over nearly three centuries, and can be summarized as follows:

### Newton's 1st Law:

Every body remains in a state of rest or uniform motion (constant velocity) unless it is acted upon by an external unbalanced force. This means that in the absence of a non-zero net force, the center of mass of a body either remains at rest, or moves at a constant speed in a straight line.

### Newton's 2nd Law:

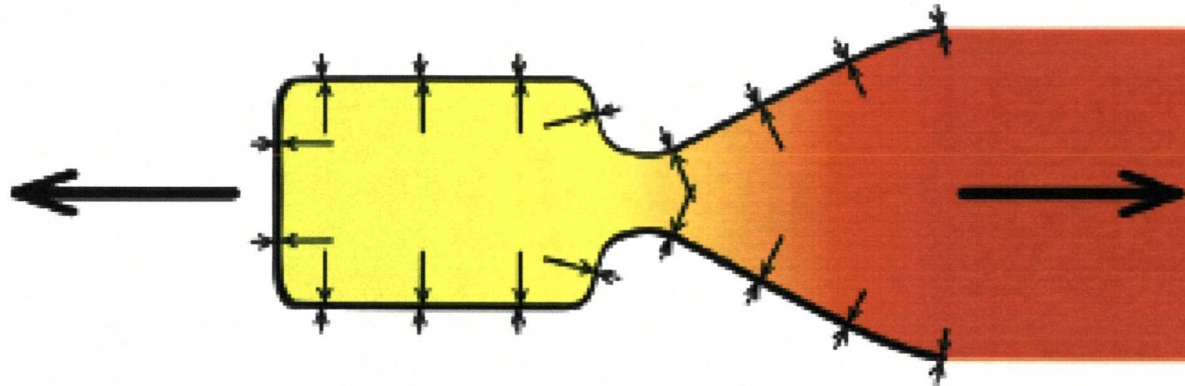
A body of mass  $m$  subject to a force  $F$  undergoes an acceleration  $a$  that has the same direction as the force and a magnitude that is directly proportional to the force and inversely proportional to the mass, i.e.,  
 $F = ma$ .

Alternatively, the total force applied on a body is equal to the time derivative of linear momentum of the body.

### Newton's 3rd Law:

The mutual forces of action and reaction between two bodies are equal, opposite and collinear. This means that whenever a first body exerts a force  $F$  on a second body, the second body exerts a force  $-F$  on the first body.  $F$  and  $-F$  are equal in magnitude and opposite in direction. This law is sometimes referred to as the action-reaction law, with  $F$  called the "action" and  $-F$  the "reaction".

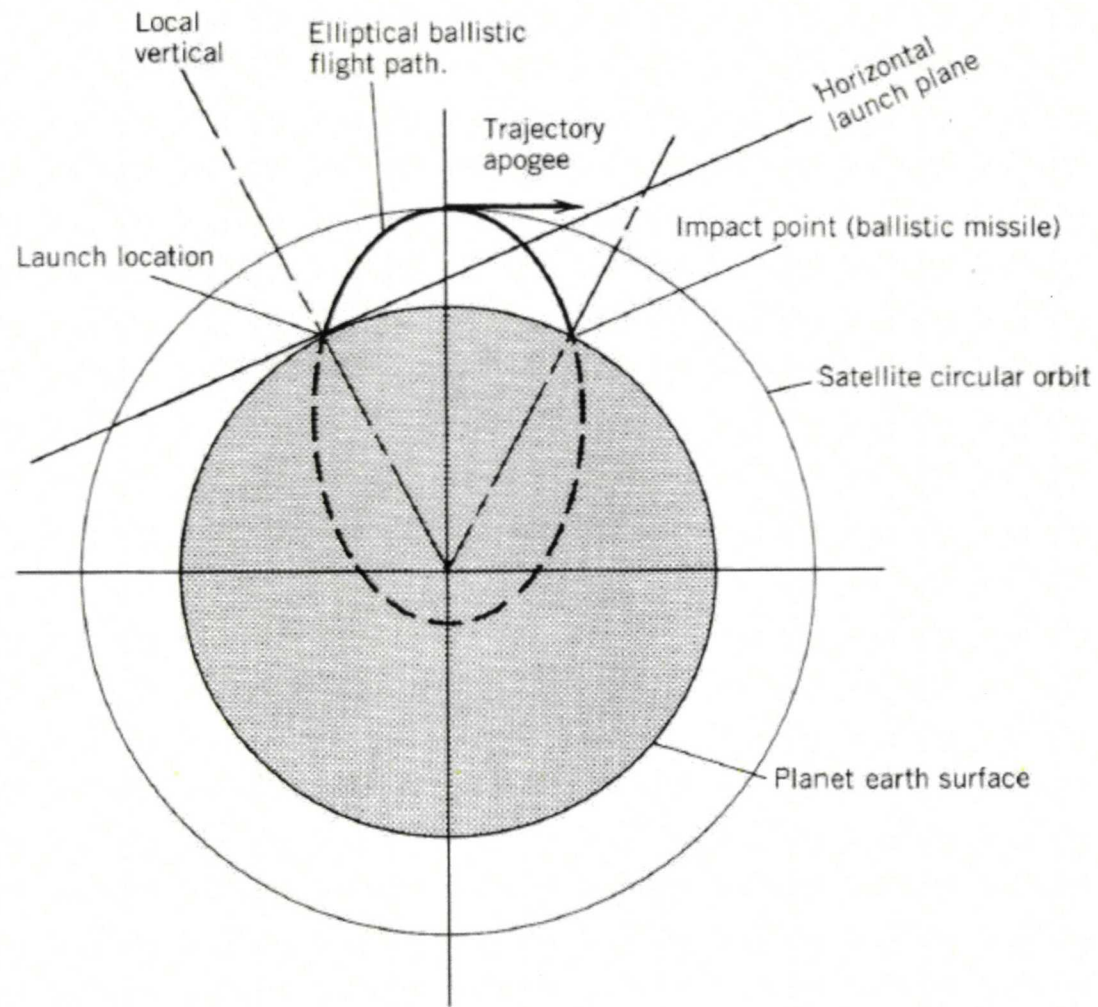
## A Simple Rocket



- Rocket engines are reaction engines and produce thrust in accordance to Newton's 3rd Law
- Rocket engines produce thrust by the expulsion of a high speed fluid (gas) exhaust created by high pressure combustion of solid or liquid propellants within a combustion chamber
- Total force applied on the vehicle is also the time derivative of linear momentum.
- Rockets can be throttled by controlling the propellant combustion rate.
- Most rockets are steered by "thrust vectoring" or in some cases, by aerodynamic surfaces (1st stage only)



## Typical Flight of a Rocket





## Rocket Science Fundamentals

---

The *Total Impulse*,  $I_t$  is the thrust force  $F$  (which can vary over time) integrated over burning time  $t$ .

$$\text{Total Impulse} \quad I_t = \int_0^t F \, dt \quad (2-1)$$

For constant thrust and negligible start/stop transients, it reduces to:

$$\text{Total Impulse} \quad I_t = Ft \quad (2-2)$$



## Rocket Science Fundamentals (Cont'd)

---

The *Specific Impulse*,  $I_s$  is the total impulse per unit weight of propellant. If the total mass flow rate is  $\dot{m}$  and standard acceleration of gravity at sea level  $g_0$  is 9.8 m/sec<sup>2</sup> or 32.174 ft/sec<sup>2</sup> then,

$$\text{Specific Impulse} \quad I_s = \frac{\int_0^t F dt}{g_0 \int \dot{m} dt} \quad (2-3)$$

This equation will give a time-averaged specific impulse value for any rocket propulsion system, particularly where the thrust varies with time. During transient conditions (during start or the thrust buildup period, or during a change of thrust levels) values of  $I_s$  can be obtained by integration or determining average values for short time intervals for  $F$  and  $m$ .

For constant thrust and propellant flow this equation can be simplified; here  $m_p$  is the total effective propellant mass.

$$\text{Specific Impulse} \quad I_s = I_t / (m_p g_0) \quad (2-4)$$



## Rocket Science Fundamentals (Cont'd)

---

For constant propellant mass flow  $\dot{m}$ , constant thrust  $F$ , and negligibly short start or short stop transients:

$$\begin{aligned} \text{Specific Impulse} \quad I_s &= F/(\dot{m}g_0) = F/\dot{w} \\ &= I_t/(m_p g_0) = I_t/w \end{aligned} \quad (2-5)$$

In an actual rocket nozzle the exhaust velocity is not uniform over the nozzle exit cross section. The velocity profile is difficult to measure accurately. For convenience of a one dimensional analysis it is assumed that the axially directed velocity is uniform over the full nozzle exit area and have a nominal value  $c$ .

This *effective exhaust velocity*  $c$  is the average equivalent velocity at which propellant is ejected from the vehicle. It is defined as

$$\text{effective exhaust velocity} \quad c = I_s g_0 = F/\dot{m} \quad (2-6)$$

$c$  is given in either meters per second or feet per second.



The *mass ratio* **MR** of a vehicle or a particular vehicle stage is defined to be the final mass  $m_f$  (after rocket operation - after propellants are consumed) divided by the initial mass  $m_o$  (before flight, when the rocket is full of fuel)

$$\text{mass ratio} \quad \mathbf{MR} = m_f/m_o \quad (2-8)$$

This applies to a single-stage or a multistage vehicle; for multistage vehicles, the overall mass ratio is the product of the individual vehicle stage mass ratios. The final mass,  $m_f$  is the mass of the vehicle after the rocket has ceased to operate and all of the useful propellant mass  $m_p$  has been consumed and ejected. The final vehicle mass  $m_f$  includes all those parts that are not propellant and can include guidance devices, navigation gear, payload, tanks, structure, residual or unused propellant and propulsion hardware. The *propellant mass fraction*  $\zeta$  indicates the percentage of propellant mass  $m_p$  in a given *vehicle* or *stage*.

*propellant mass fraction*

$$\zeta = m_p/m_o = (m_o - m_f)/m_o = m_p/(m_p + m_f) \quad (2-9)$$

$$m_o = m_f + m_p \quad (2-10)$$

The *impulse-to-weight* ratio of a complete propulsion system is defined as the total impulse,  $I_t$  divided by the initial vehicle weight or loaded vehicle weight  $w_0$  (loaded with propellants). A high value indicates an efficient design. For a flight with a constant value of  $\dot{m}$  or  $I_s$ , it can be expressed as

$$\text{impulse-to-weight} \quad \frac{I_t}{w_0} = \frac{I_s t}{(m_f + m_p)g_0} = \frac{I_s}{m_f g_0/t + \dot{m} g_0} \quad (2-11)$$

It can be shown that as the fraction of propellant becomes very large ( $\zeta$  approaches 1.0), the value of the impulse-to-thrust ratio approaches that of the specific impulse. The value of  $I_t/w_0$  cannot exceed the value of  $I_s$ .

The *loaded propulsion system mass* usually consists of propulsion system hardware mass (hardware necessary to store and burn propellant) and the propellant mass. It is the vehicle mass diminished by those items that are not aerodynamic fins, nose cone, payload, and navigation system.

**The *thrust-to-weight ratio* expresses the acceleration (in multiples of the acceleration of gravity) that the engine is capable of giving to its own payload propulsion system mass.**

The weight is the mass multiplied by the sea level acceleration of gravity or 9.8066 m/sec<sup>2</sup> or 32.174 ft/sec<sup>2</sup>.



## Thrust

---

The thrust force of a rocket is the reaction experienced by its structure due to the ejection of high-velocity matter. Like water pushes back a garden hose and shooting a gun causes it to recoil. In the latter case, the forward momentum of the bullet and the powder charge is equal to the recoil or rearward momentum of the gun barrel.

*Momentum* is defined as the product of mass and velocity,  $M = mv$

In rocket propulsion only relatively small gas masses are used, which are carried within the vehicle and ejected at a very high velocity. The efflux of hot gases from a rocket vehicle can basically be regarded as the ejection of small masses such as  $\Delta m$  (the molecules of the gas) at a high relative velocity  $v_2$  with respect to the vehicle, which has a mass  $m_v$  and is moving at the vehicle velocity  $u$ .

**The force ( $F$ ), the momentum ( $M$ ), and the velocity( $u$  &  $v$ ) are vector quantities.**

## Thrust (Cont'd)

Consider a simplified system where only one particle  $\Delta m$  is being ejected at any one time. The net momentum gained by the vehicle has to equal the momentum of the ejected gas.

$$m_v \Delta u = \Delta m(u - v_2)$$

Differentiating with respect to time, one obtains  $dM/dt = 0$ ; because no external momentum change is applied. Then  $d(\Delta m)/dt$  approaches  $-dm/dt$  for a continuously flowing gas. This equals the rate of decrease of the vehicle mass  $m_v$ . If the vehicle velocity  $u$  is constant, its time derivative is equal to 0. Also,  $\Delta m$  and  $\Delta u$  are very small and approach 0 as their limit. Thus one obtains

$$m_v \frac{du}{dt} = -\frac{dm}{dt} v_2 \quad (2-12)$$

The left hand term is defined by Newton's 2<sup>nd</sup> Law as being equal to the thrust force  $F$ . For the case where the thrust  $F$  and flow  $\dot{m}$  are constant,

$$F = \frac{dm}{dt} v_2 = \dot{m} v_2 = \frac{\dot{w}}{g_0} v_2 \quad (2-13)$$



## Thrust (Cont'd)

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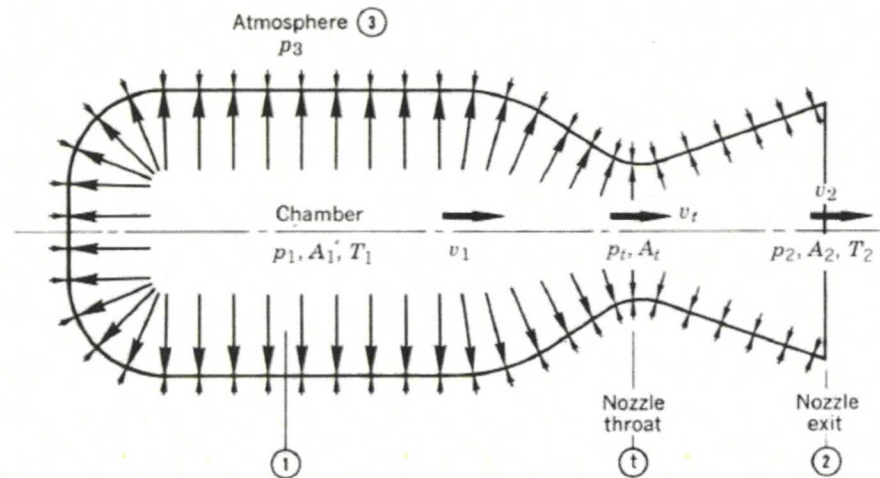
$$F = \frac{dm}{dt} v_2 = \dot{m} v_2 = \frac{\dot{w}}{g_0} v_2 \quad (2-13)$$

This force is the thrust obtained for any true rocket propulsion system where the nozzle exit pressure is equal to the atmospheric pressure. It assumes a uniform axial exhaust velocity that does not vary across the area of the jet. The minus sign in equation 2-12 indicates a decrease in the mass of the vehicle, or it can be interpreted that the velocity  $v_2$  is in the opposite direction to the thrust  $F$ . In most rockets the propellant mass flow rate does not vary, so  $\dot{m}$  is used instead of  $-dm/dt$ ; and the weight flow rate  $\dot{w}$ , usually a measured quantity, when divided by the acceleration of gravity  $g_0$ , again represents mass flow rate.

**The preceding equation shows that the thrust is proportional to the propellant flow rate and the exhaust velocity.**

## Thrust (Cont'd)

The pressure of the surrounding fluid (usually air) has an influence on the thrust. The following figure shows schematically the external pressure acting uniformly on the outer surface of a rocket chamber and the gas pressures on the inside on a typical rocket engine. The size of the arrows the relative magnitudes of the pressure forces. The axial thrust can be determined by integrating all the pressures acting on the areas that can be projected on a plane normal to the nozzle axis. The radially outward acting forces are appreciable but do not contribute to the axial thrust, because the rocket is axially symmetrical. By inspection it can be seen (and it can be proven) that at the exit area  $A_2$  of the engine's gas exhaust there is an unbalance of the external environmental or atmospheric pressure  $p_3$  and the local pressure  $p_2$  of the hot gas jet at the exit plane of the nozzle



For a steadily operating rocket propulsion system flying in a homogeneous atmosphere (neglecting localized boundary layer and drag effects), the thrust is equal to

$$F = \dot{m}v_2 + (p_2 - p_3)A_2 \quad (2-14)$$



## Thrust (Cont'd)

---

$$F = \dot{m}v_2 + (p_2 - p_3)A_2 \quad (2-14)$$

The thrust acting on the vehicle is composed of two terms. The first term, the *momentum thrust*, is the product of the propellant mass flow rate and the exhaust velocity relative to the vehicle. The second term, the pressure thrust, consists of the product of the cross-sectional area of the nozzle exit (where the exhaust jet leaves the vehicle) and the difference between the exhaust pressure and the ambient fluid pressure. If the exhaust pressure is less than atmospheric pressure, the pressure thrust is negative. Because this condition gives a low thrust and is undesirable, the rocket exhaust nozzle is usually so designed that the exhaust pressure is equal to or slightly higher than atmospheric pressure. When the fluid pressure is equal to the exhaust pressure, the pressure thrust term is zero, and the thrust is the same as (see Equation 2-13)

$$F = (\dot{w}/g_0)v_2 = \dot{w}I_s = \dot{m}v_2 \quad (2-15)$$

The rocket nozzle design, which permits the expansion of the propellant products to the pressure that is exactly equal to the pressure of the surrounding fluid (atmospheric air), is referred to as the rocket nozzle with *optimum expansion ratio*.

*Example 2-1.* A rocket projectile has the following characteristics:

Initial mass	200 kg
Mass after rocket operation	130 kg
Payload, nonpropulsive structure, etc.	110 kg
Rocket operating duration	3.0 sec
Average specific impulse of propellant	240 N-sec <sup>3</sup> /kg-m

Determine mass ratio, propellant mass fraction, propellant flow rate, thrust, thrust-to-weight ratio, acceleration of vehicle, effective exhaust velocity, total impulse, and the impulse-to-weight ratio.



## Rocket Science Fundamentals (Cont'd)

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SOLUTION. Mass ratio of vehicle (Equation 2-8)  $MR = m_f/m_0 = 130/200 = 0.65$ ; mass ratio of rocket system  $MR = m_f/m_0 = (130 - 110)/(200 - 110) = 0.222$ . Note that the empty and initial masses of the propulsion system are 20 and 90 kg, respectively.

The propellant mass fraction (Equation 2-9) is

$$\zeta = (m_0 - m_f)/m_0 = (90 - 20)/90 = 0.778$$

The propellant mass is  $200 - 130 = 70$  kg. The propellant mass flow rate is  $\dot{m} = 70/3 = 23.3$  kg/sec.

The thrust (Equation 2-5) is

$$F = I_s \dot{w} = 240 \times 23.3 \times 9.80 = 54,800 \text{ N}$$

The thrust-to-weight ratio of the vehicle is

$$\text{initial value } F/w_0 = 54,800/(200 \times 9.80) = 28$$

$$\text{final value } 54,800/(130 \times 9.80) = 43$$

The maximum acceleration of the vehicle is  $43 \times 9.80 = 421 \text{ m/sec}^2$ . The effective exhaust velocity (Equation 2-6) is

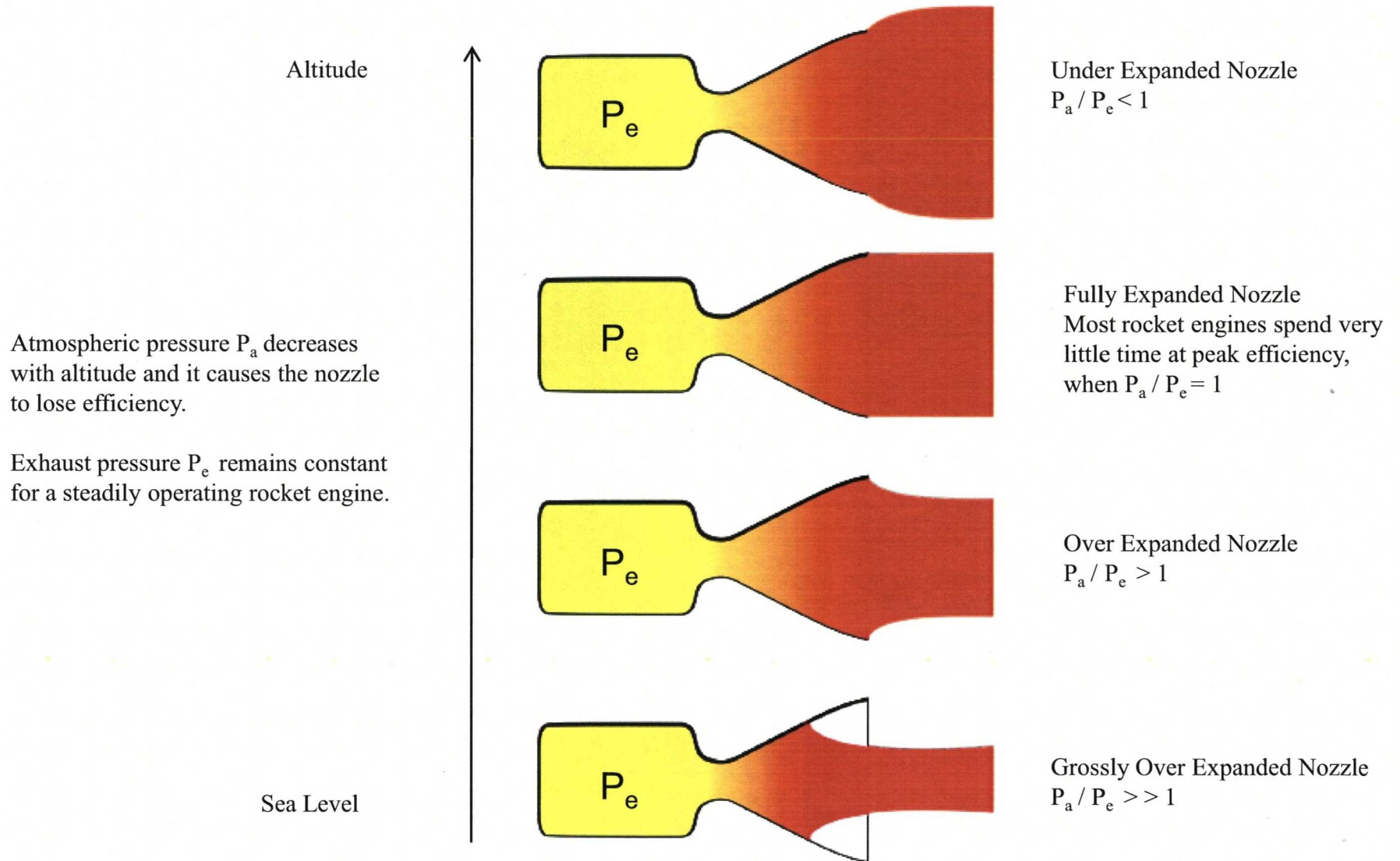
$$c = I_s g_0 = 240 \times 9.80 = 2352 \text{ m/sec}$$

The total impulse (Equations 2-2 and 2-5) is

$$I_t = I_s w = 240 \times 70 \times 9.80 = 164,600 \text{ N-sec}$$

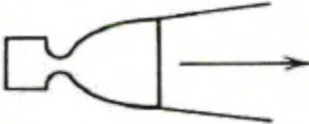

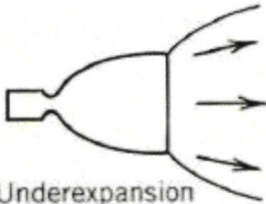
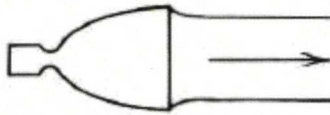
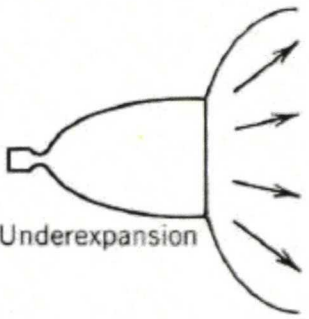
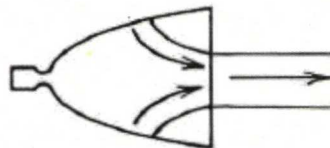
This result can also be obtained by multiplying the thrust by the duration. The impulse-to-weight ratio (Equation 2-9) is  $I_t/w_0 = 54,870/[(200 - 110)9.80] = 187$ .

## Effect of Atmospheric Pressure

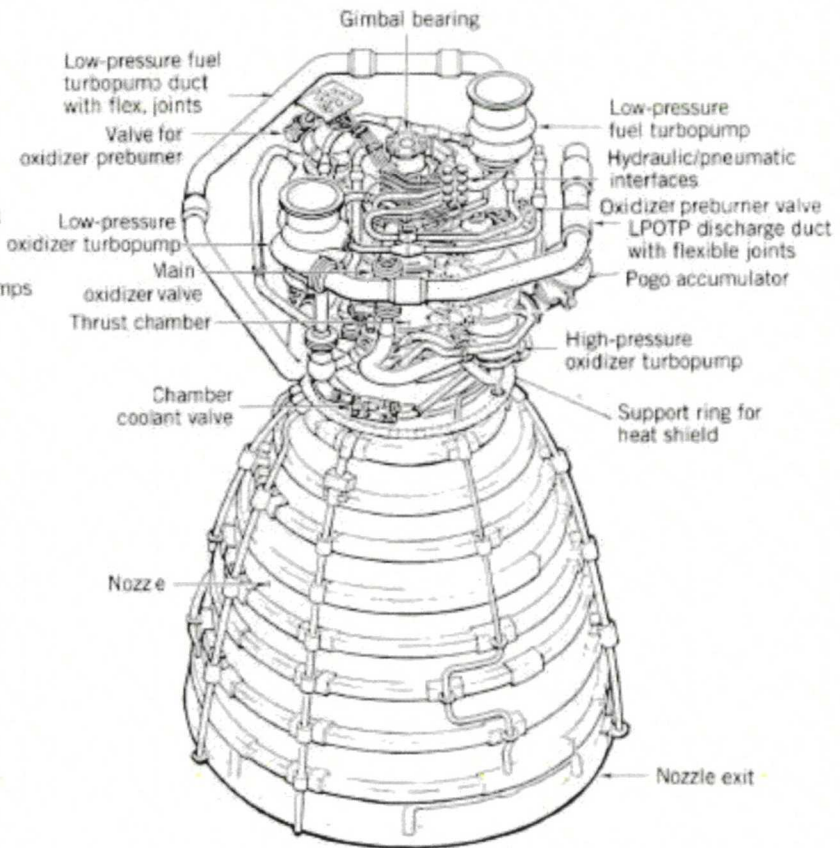
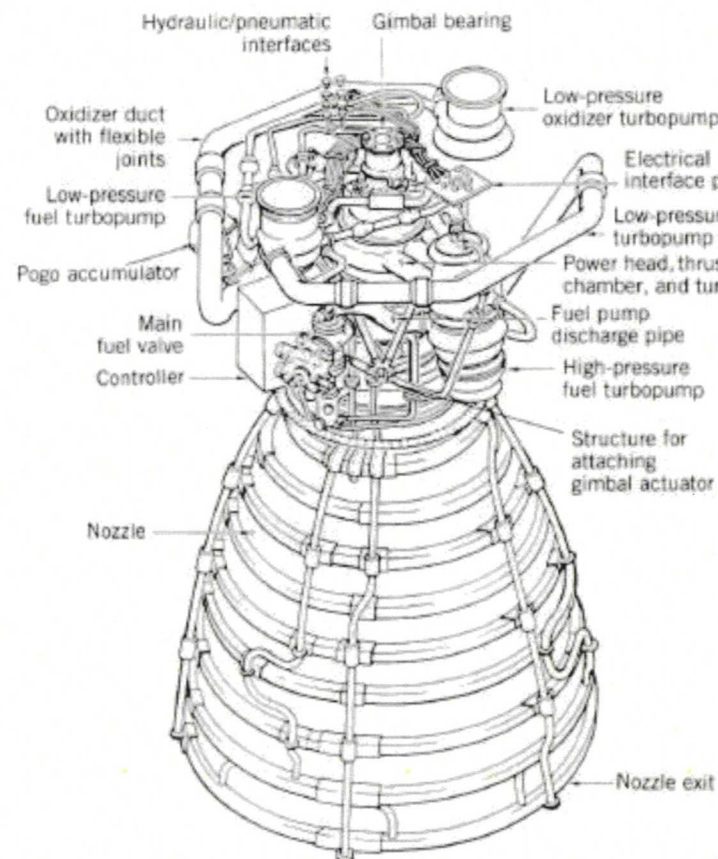




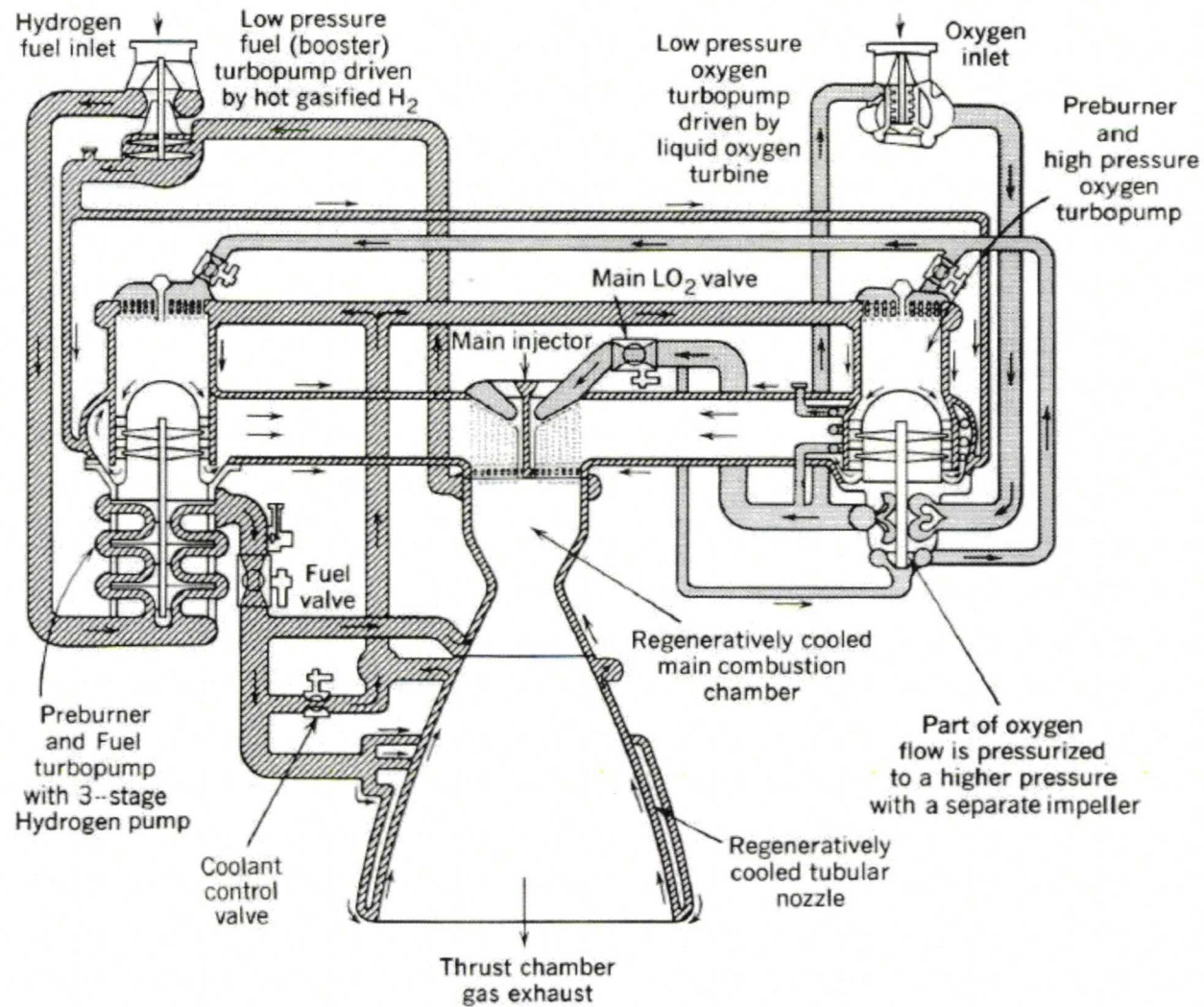
## Effect of Atmospheric Pressure

Stage	$A_2/A_t$	During flight		During sealevel static tests	
			$h(\text{km}) \quad I_s (\text{sec})$		$h(\text{km}) \quad I_s (\text{sec})$
Booster or first stage	6	Nozzle flows full slight underexpansion	0      267	Nozzle flows full	0      267
Second stage	10		24      312		0      254
		Underexpansion		Overexpansion, slight contraction	
Third stage	40		100      334		0      245
		Underexpansion		Flow separation caused by overexpansion	

## Two views of Space Shuttle Main Engine (SSME)

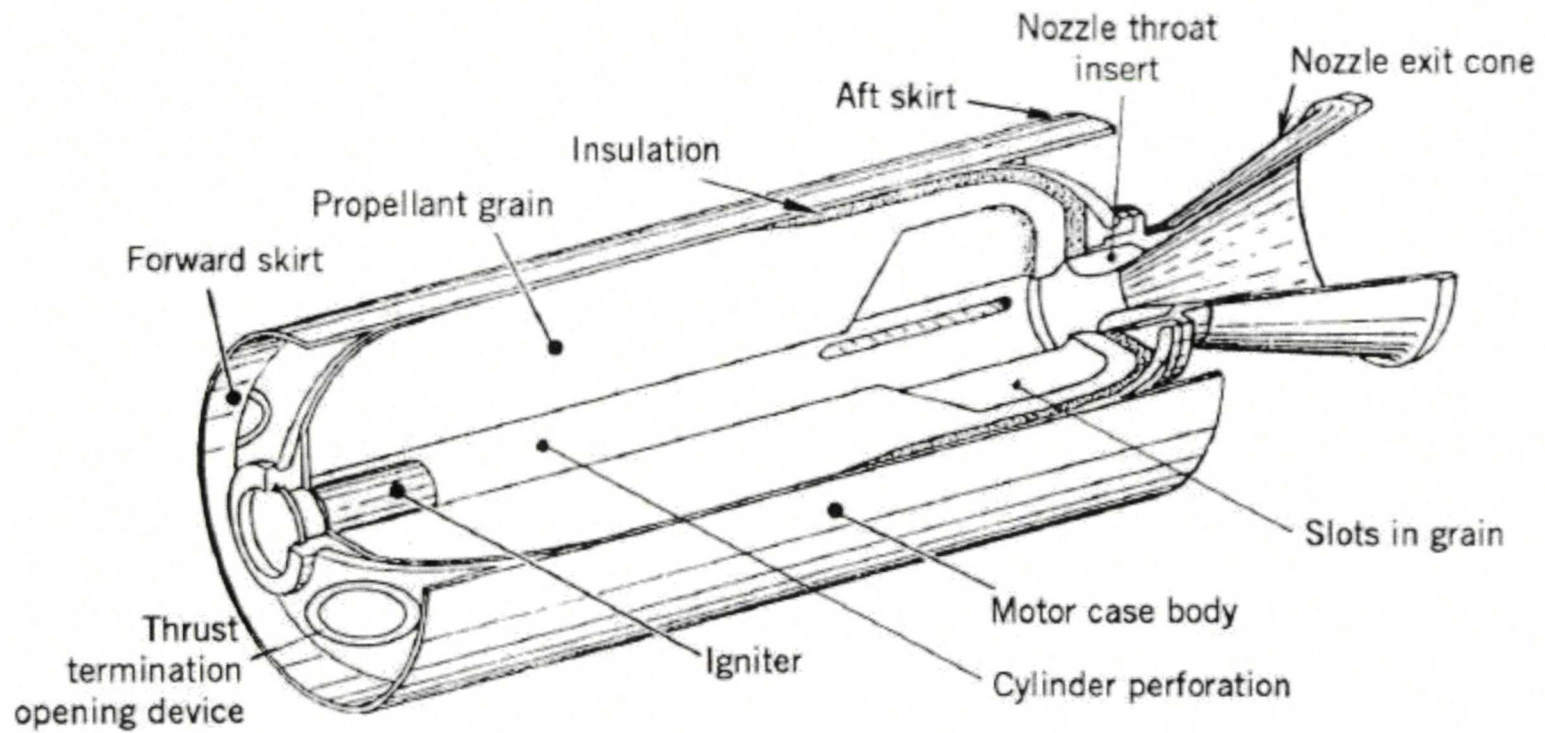


## Propellant & Oxidizer Flow Diagram of SSME

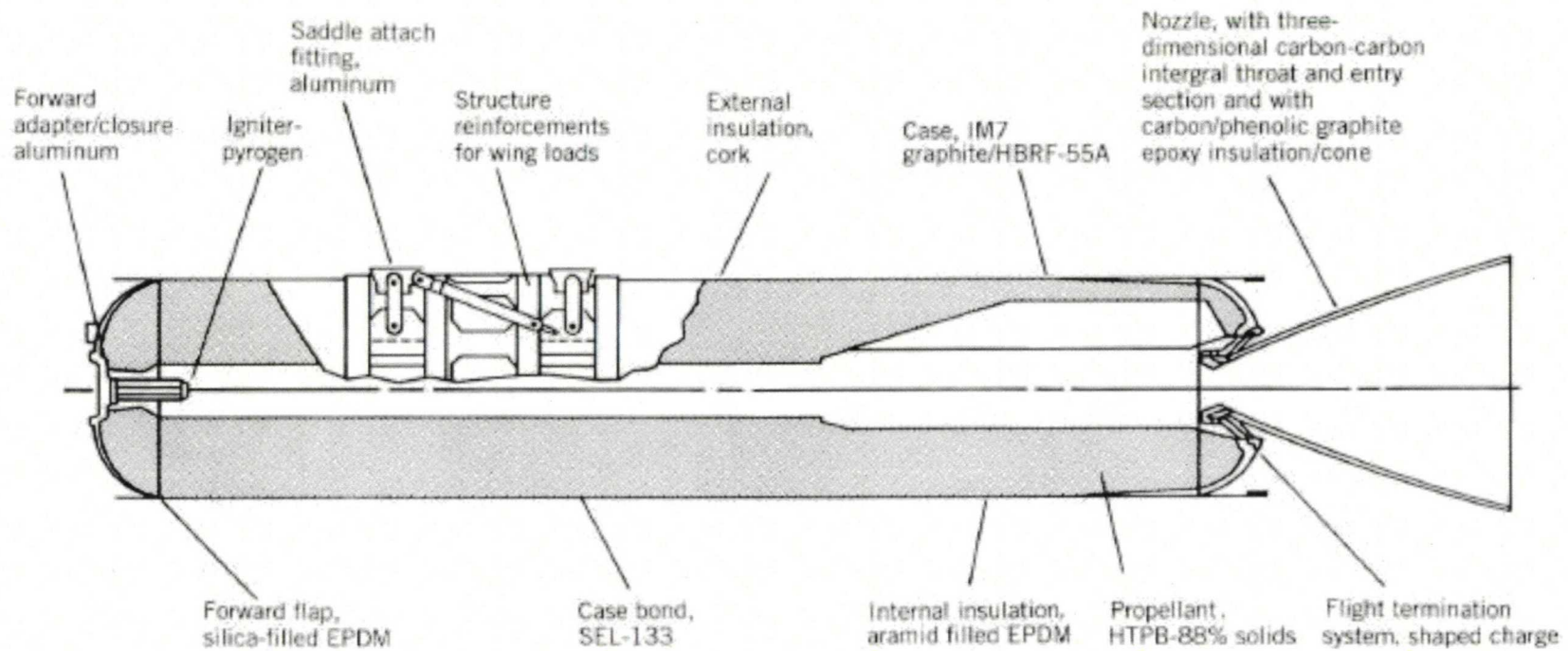




## Typical Solid Rocket Motor



## Pegasus First Stage Motor



## Examples of *Specific Impulse* ( $I_s$ )

Propellant Mix	Vacuum $I_s$ (sec)	Effective Exhaust Velocity $c$ (meters/sec)
LH <sub>2</sub> /LOX	455	4462
Kerosene (RP-1)/LOX	358	3510
Hydrazine/N <sub>2</sub> O <sub>4</sub>	305	2993
Solid Motor	250	2500

While *Specific Impulse* is the key metric in rocket propulsion efficiency, there are other important considerations that influence the design of a rocket powered vehicle.

For instance:

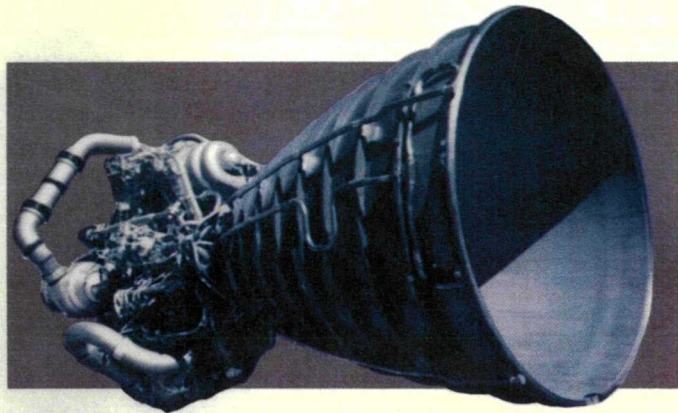
- LH<sub>2</sub>/LOX/ is absolutely clean burning but has low propellant density which drives up the size of the fuel tank and turbo machinery and therefore the vehicle. Requires insulated tanks to minimize boil off and continuous replenishing to keep tanks full before launch.
- Solid Motors can be stored for years after casting but must be shipped loaded (heavy and hazardous) across the country and the exhaust is environmentally damaging.
- Hydrazine and N<sub>2</sub>O<sub>4</sub> are storable at room temperature but extremely toxic and environmentally damaging.
- RP-1/LOX is arguably the best choice for a 1st stage .
- Using strap-on solid motors to augment the 1<sup>st</sup> Stage provides additional flexibility to mission design.



# ROCKET PROPULSION ELEMENTS

An Introduction to the  
Engineering of Rockets

Sixth Edition



**GEORGE P. SUTTON**

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